TOC

Chapter 6

SPACECRAFT DESIGN, STRUCTURE AND OPERATIONS

A typical spacecraft consists of a mission payload and the bus (or platform). The bus is made up of five supporting subsystems: structures, thermal control, electrical power, attitude control and telemetry, tracking and commanding (TT&C).

SPACECRAFT DESIGN PROCESS

The design and development of a spacecraft is an engineering process divided into identifiable phases. In the past, the spacecraft development process required several years to accomplish, especially if it was a government project. Recent advances in satellite manufacturing technology, coupled with the growth of commercial markets in satellite communications and imaging, have significantly shortened the process. This change is described by a former commander of U.S. Air Force Space Command:

> "…new commercial geosynchronous satellites are available 18 months after order—soon to be down to 12 months. For the small LEO systems, time from order to delivery is about three years...In contrast, the acquisition of national security systems runs 10 to 15 years... the same plant will build three hundred Teledesic satellites in three years and 15 Global Positioning System (GPS) satellites in seven years..."
>
> - General Thomas Moorman Jr., USAF, Ret. in *Air Power Journal*, Spring 1999

Regardless of how long it takes, spacecraft design and development typically occur in phases identified as: requirements definition, conceptual design, preliminary design, and detailed (or critical) design. These phases and their sequential relationships are shown in **Figure 6-1**.

Spacecraft development begins by defining needs or requirements the system is to satisfy. For example, the need to gather and store weather images and data. Or, take photographic images with less than 15 meter resolution and transmit the information in real time. The spacecraft mission will be a major determiner of the type orbit chosen for the space craft.

Next is the conceptual design phase, in which various system concepts which can satisfy the mission requirements are considered and subjected to analysis. The most proficient means to carry out the mission is selected and major risks, costs, and schedules are identified.

The preliminary design phase follows conceptual design, and may stretch over a couple of years **(Fig. 6-2)**. During this phase, variations of the concept chosen in the conceptual design phase are analyzed and refined. Subsystem and component level specifications are defined and major documents such as the interface control document are written. Anticipated performance of systems and subsystems is substantiated and, from the detailed specifications, a preliminary parts list is identified.

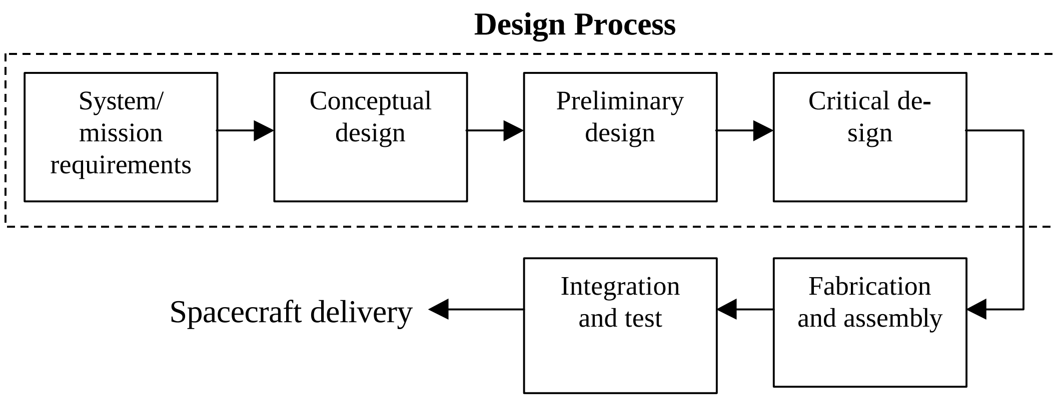

Fig. 6-1 Spacecraft design phases

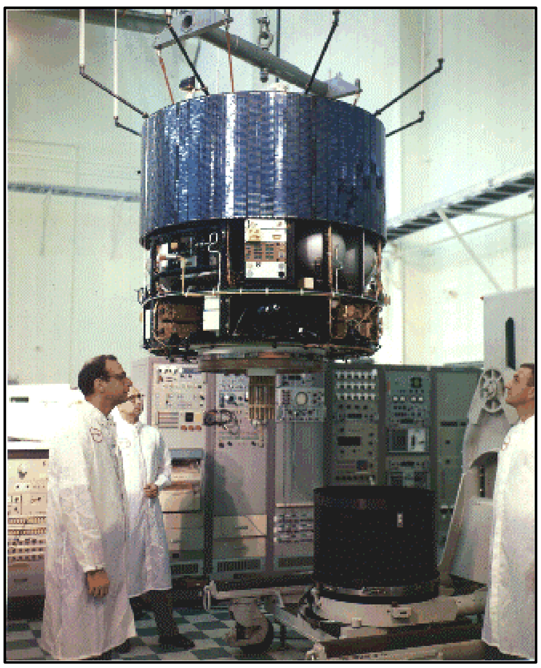

Fig. 6-2 Spacecraft in Preliminary Design

The final phase is the critical (or detailed) design phase which can take four to five years. It is within this phase that the specific aspects of structural design are identified, such as finalizing the thickness of structures and load paths. The spacecraft design must accommodate everything that fits into the structure, including equipment, crew, provisions and payload. Options for secondary structures (plumbing, wiring, etc.) are also analyzed and evaluated several times during the critical design phase. Design verification is an important part of this phase. Verification involves tests of electronic circuit models, software and engineering models. Design and performance margin estimates are refined and test and evaluation plans finalized.

STRUCTURES SUBSYSTEM

The functions of the structural subsystem are to enclose, protect and support the other spacecraft subsystems and to provide a mechanical interface with the launch vehicle.

The enclose and protect functions are especially necessary during spacecraft assembly, handling and transportation from the manufacturing facility to the launch facility. Structural members provide the mating and attachment points for subsystem components such as batteries, propellant tanks, electronics modules and so on. The structure must also sustain the stresses and loads experienced during environmental testing, launch, perigee and apogee firings, and deployment of booms, solar arrays and antennas. Noises and vibrations can be especially severe as the spacecraft experiences high G forces during launch. Acoustic noise is the highest in the early stages of the launch and is transmitted from the rocket motors by the air through the fairings or housing and into the spacecraft. Steady loads are transmitted through the structure as the rockets accelerate the spacecraft to the high velocities required for injection into orbit.

A wide range of vibration frequencies is transmitted through the spacecraft supports from the rocket motors. The separation ring and other pyrotechnic devices send sudden shocks through the structure.

Upon reaching its final orbit position, the loads on the spacecraft are greatly reduced in the zero gravity environment, but the alignment requirements are more rigorous. The designer must satisfy all requirements, minimize the structure mass and cost, and still keep the probability of failure near zero.

Structure Types

There are two main types of satellite structures: open truss and body mounted. An open truss structure has a specific shape to it (see **Fig. 6-3**), usually a box or a cylinder.

Inside the body of the spacecraft is a honeycomb structure where the equipment boxes are attached. In a body mounted structure equipment is attached directly on primary equipment such as an antenna or apogee kick motor. These satellites do not have a specific shape to them. There are also combinations of these two structure types where part of the satellite would have a shape such as a box, with some equipment attached to the exterior.

Materials

When designing a component for structural use in a spacecraft, the engineer must at some point in the analysis, decide what materials to use (**Fig. 6-4**). Thousands of different materials are used in making a spacecraft. Many of them serve dual or triple roles to save weight and avoid complexity. For example, the frame of a spacecraft could be a heat sink and electrical ground as well as the main structure.

During its lifetime, the spacecraft will be subjected to severe conditions. These may include various mechanical loads, vibrations, thermal shocks, electrical charges, radiation, and a chemical and particulate environment that starts with the salt and sand spray at our launch sites.

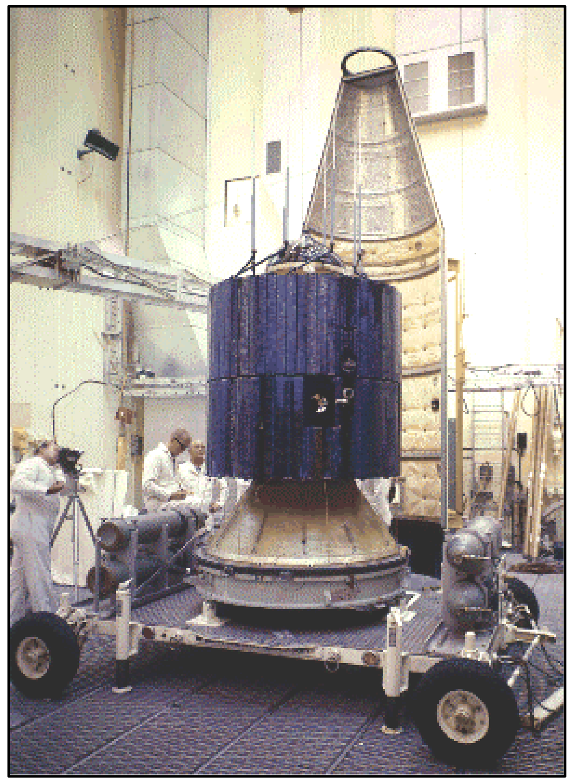

Fig. 6-3 Satellite being fitted to rocket faring

The material selected must meet standards of yield, ultimate and fatigue strength, specific stiffness, hardness and toughness, ductility, thermal expansion, creep resistance and melting point. Other properties must also be examined, since structural materials often serve more than one role. Depending on the circumstances, electrical conductance may be a favorable attribute of a structural member. Thermal conductance and capacitance may also be desirable depending upon whether the structure is to insulate internal components or relieve thermal stresses by conducting thermal energy away from hot spots.

Fig. 6-4 Various materials evident in a spacecraft

Finally, the availability, formability and ease with which parts can be machined out of a particular material will influence the selection process. Some are scarce and expensive. Others are extremely brittle or soft. Some are hard to cast, forge or to machine. Almost every material presents some type of fabrication problem.

Aluminum, magnesium, titanium and beryllium are the elements that make up the major lightweight alloys used in space vehicles. They are all much lighter than steel and are non-magnetic. Aluminum alloys are the most widely used structural materials. Their strength to weight ratio is equal to steel, which combined with its availability and ease of manufacture, makes them very desirable. Aluminum can be made into wire, thin sheets, cast into complex and thick parts, machined, welded, forged and stamped. Unfortunately, aluminum and its alloys are unable to withstand high temperatures. Prolonged exposure to temperatures above 400° F results in a loss of strength and creep can set in at lower temperatures.

Magnesium is lighter than aluminum, but not as strong. It is useful for lower strength, lightweight applications at temperatures up to 400° F. Fabrication is similar to that of aluminum in that parts can be made and joined together in much the same way. Corrosion in the presence of moisture is a problem with magnesium and its alloys; coatings and finishes are needed for protection.

Titanium can replace aluminum in higher temperature environments, as it has the ability to remain strong at temperatures up to 1,200° F. In situations where a structure must be lightweight and strong when subjected to 400 - 1,200° F, aluminum cannot be used. Unfortunately, titanium is not as light or durable as aluminum. It has a tendency to become brittle at low temperatures and when placed under repeated loads. It is also more difficult to weld.

Beryllium is used to make phenomenally light alloys. Its strength is close to that of steel and its density is comparable to aluminum. This makes for extremely stiff, lightweight structures. An added plus is its ability to retain its properties at temperatures up to 1,000° F.

Beryllium is more difficult to fabricate than aluminum and is susceptible to surface damage while it is being machined due to its brittle properties. An additional consideration is beryllium's toxicity. It presents a serious health hazard to unprotected workers. Finally, beryllium is more costly than many other metals.

Graphite, plastics, nylon and ceramics comprise the non-metallic materials used in spacecraft. Graphite is not usually thought of as a structural material. It is weak and brittle at room temperature, but is widely used as a thermal protection material. Since the strength of graphite improves with temperatures up to about 4,500° F, it is very possible that vehicles which must enter Jupiter's atmosphere, or orbit very close to the Sun, may have some structural parts made out of graphite.

Plastic has many desirable qualities as a spacecraft component material. It is very inexpensive, very available and easy to fabricate into intricate shapes. It is also durable and is a good electrical and thermal insulator. For spacecraft interiors, where temperatures are relatively low, plastic may be a good replacement for light alloys.

Nylon has a unique advantage in that mechanisms made of it may not need lubrication. Nylon may be the optimal material for low power gear trains in space.

The general property of ceramics is that they are extremely weak in tension and are very brittle. They can, however, withstand very high temperatures, protecting themselves by gradual erosion. Hence, ceramics are useful in some radomes, jet vanes, leading edges and solid rocket nozzles.

In the future, aerospace fabrication will make greater use of composites. Composites are two or more materials manufactured together to form a single piece that can have almost any property an engineer specifies. Uni- or omni-directional strength, resistance to high temperatures and resistance to corrosives are a few of these properties. Examples of composites are: fiberglass and carbon epoxy, both structural materials; and carbon-composite, a thermal protection material used on leading edges of the space shuttle.

THERMAL CONTROL SUBSYSTEM

The spacecraft thermal environment is determined by the magnitude and distribution of radiation from the Sun and Earth. The purpose of the spacecraft thermal control subsystem is to control the temperature of individual components to ensure proper operation through the life of the mission. Some components are required to be maintained below a critical temperature, i.e. high temperature limits the reliability and lifetime of transistors due to increased electromigration effects. Optical sensors require temperature be maintained within a critical range to minimize lens distortion, and hydrazine propellant must be maintained above a critical temperature (10° C) or it will freeze.

The thermal control process has to meet the requirements of all subsystems. Balance between structural and thermal requirements is necessary to achieve the best spacecraft configuration to permit proper thermal balance.

The thermal control subsystem uses every practical means available to regulate the temperature on board a satellite. Selection of the proper thermal control system requires knowledge of mission requirements as well as the operational environment. Temperatures within space vehicles are affected by both internal and external heat sources.

Sources of Thermal Energy

The sources of heat energy in a spacecraft include people (in manned missions), electronic equipment, frictional heat generated as the vehicle leaves or reenters the atmosphere, the Sun, heat reflected from the Earth (altitude dependent), and Earth thermal radiation (altitude dependent). Thermal control techniques can be divided into two classes: passive thermal control and active thermal control.

Passive Thermal Control

A passive thermal control system maintains temperatures within the desired temperature range by control of the conductive and the radiative heat paths. This is accomplished through the selection of the geometrical configuration and thermo-optical properties of the surfaces. Such a system does not have moving parts, moving fluids or require electrical power. Passive system offer the advantages of high reliability due to absence of moving parts or fluid, effectiveness over wide temperature ranges and light weight. A disadvantage is low thermal capacity. Passive thermal control techniques include thermal coatings, thermal insulations, heat sinks and phase change materials.

Spacecraft external surfaces radiate energy to space. Because these surfaces are also exposed to external sources of energy, their radiative properties must be selected to achieve a balance between internally dissipated energy, external sources of energy and the heat rejected into space. The two properties of primary importance are the emittance of the surface and solar absorbency. Paints and coatings can be used to reduce reflection and to increase or decrease absorption of heat or light energy.

Two or more coatings can be combined in an appropriate pattern to obtain a desired average value of solar absorbence and emittance (i.e. a checkerboard pattern of white paint and polished metal).

For a radiator, low absorbence and high emittance are desirable to minimize solar input and maximize heat rejection to space. For a radiator coating, the initial values are important because of degradation over the lifetime of the mission. Degradation can be significant for all white paints. For this reason, the use of a second surface mirror coating system is preferred. An example of such a coating is vapor deposited silver on 0.2 mm thick fused silica, creating an optical solar reflector. Degradation of thermal coating in the space environment results from the combined effects of high vacuum, charged particles and ultraviolet radiation from the Sun.

Thermal insulation is designed to reduce the rate of heat flow per unit area between two boundary surfaces at specified temperatures. Insulation may be a single homogeneous material such as low thermal conductivity foam or an evacuated multi-layer insulation in which each layer acts as a low-emittance radiation shield and is separated by low-conductance spacers.

Multi-layer insulations are widely used in the thermal control of spacecraft and components in order to:

- Minimize heat flow to or from the component
- Reduce the amplitude of temperature fluctuations in components due to time-varying external radiative heat flux
- Minimize the temperature gradients in components caused by varying directions of incoming external radiative heat.

Multi-layer insulation consists of several layers of closely spaced radiation-reflecting shields which are placed perpendicular to the heat-flow direction. The aim of the radiation shields is to reflect a large percentage of the radiation the layer receives from warmer surfaces. Heat sinks are materials of large thermal capacity, placed in thermal contact with the components whose temperature is to be controlled. When heat is generated by the component, the temperature rise is restricted because the heat is conducted into the sink. The sink will then dispose of this heat to adjacent locations through conduction or radiation (**Fig. 6-5**). Heat sinks are commonly used to control the temperature of those items of electronic equipment which have high dissipation, or a cyclical variation in power dissipation.

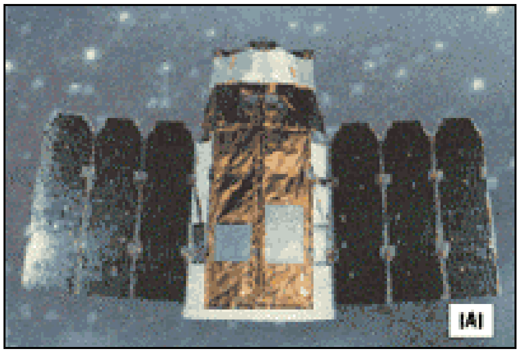

Fig. 6-5. Satellite body with thermal heat sinks

Solid-liquid phase-change materials (PCM) present an attractive approach to spacecraft passive thermal control when the incident orbital heat fluxes, or on-board equipment dissipation, changes widely for short periods. The PCM thermal control system consists primarily of a container filled with a material capable of undergoing a chemical phase change. When the temperature of spacecraft surfaces increases, the PCM will absorb excess heat through melting. When the temperature decreases, the PCM gives heat back and solidifies. Phase-change materials used for temperature control are those with melting points close to the desired temperature of the equipment. Then the latent heat associated with the phase change provides a large thermal inertia as the temperature of the equipment passes through the melting point. However, the phase-change material cannot prevent a further temperature rise when all the material is melted.

One of the more common methods of rejecting electronic heat is to mount the electronics just inside the spacecraft bus structure. Thus, the energy is conducted over a short path to an external spacecraft thermal control surface (frequently referred to as a radiator and sometimes as a shearplate). This surface is usually coated with a low solar absorptance/high infrared emittance coating (usually a white paint). Such surfaces are usually positioned by spacecraft orientation to point to deep space. Thus, the natural environment is minimized or eliminated and maximum heat rejection occurs.

Active Thermal Control

Passive thermal control may not be adequate and efficient for the applications where the equipment has a narrowly specified temperature range, or where there is great variation in equipment power dissipation and solar flux during the mission. In such cases, temperature sensors may be placed at critical equipment locations. When critical temperatures are reached, mechanical devices are actuated to modify the thermo-optical properties of surfaces, or electrical power heaters turn on or off to compensate for variations in the equipment power dissipation. Active thermal control techniques include louvers, electrical heaters, refrigerative thermal control and expendable heat sinks.

For a spacecraft in which the changes in internal power dissipation or external heat fluxes are severe, it is not possible to maintain the spacecraft equipment temperatures within the allowable design temperature limits unless the ratio of absorbence to emissivity can be varied. A very popular and reliable method which effectively gives a variable ratio is through the use of louvers. When the louver blades are open, the effective ratio is low (low absorbtivity, high emissivity); when the blades are closed, the effective ratio is high (high absorbtivity, low emissivity). The louvers also reduce the dependence of spacecraft temperatures on the variation of the thermo-optical properties of the radiator.

Louvers consist of five main components: baseplate, blades, actuators, sensing elements and structural elements. The baseplate is a surface of low absorbence to emittance ratio which covers the critical set of equipment whose temperature is being controlled. The blades, driven by actuators, are the elements of the louvers that give variable radiation characteristics at the baseplate. When the blades are closed, they shield the baseplate from its surroundings. When they are fully open, the coupling by radiation from the baseplate to the surroundings is the largest. The radiation characteristics of the baseplate can be varied in the range defined by these two extreme positions of the blades. The actuators are the elements of the louvers which drive the blades according to the temperature sensed by sensors placed in the baseplate. The commonly used actuators are bimetal springs or bellows. Generally, bimetals are used with the multiple-blade actuation system and bellows with the single-blade actuation system.

Electrical heaters (resistance elements) are used to maintain temperatures above minimum allowable levels. The heater is typically part of a closed loop system that includes a temperature sensing element and an electronic temperature controller (thermostat). Electrical heaters are used in an on-off control mode, a ground controllable mode, a proportional control mode or simply in a continuous-on mode. The heaters are strips of kapton with etched foil-heating elements and welded power leads. Heaters are bonded on structures to maintain temperature levels and gradients consistent with interface and alignment requirements. In all applications, primary and backup redundant sets of heaters should be implemented and controlled by redundant mechanical thermostats with predetermined set-points.

Some sensors require constant cold temperatures. These types of sensors on board spacecraft must be isolated from other system components and may need a cooling system to function properly. A closed system refrigeration cycle may be necessary for high heat loads.

Radiators are closed loop systems used in conjunction with other types of thermal control devices. They are active due to this interaction (i.e., use working fluids etc.). Radiator systems require large surface areas to dissipate heat into space; a major disadvantage of this type of system.

Expendable heat sinks work by transferring heat to a fluid or gas and then the fluid or gas is vented overboard. Thus, the working fluid or gas is expended. Water, because of its high latent heat of vaporization, is generally the best expendable coolant. This is an open loop system.

ELECTRICAL POWER SUBSYSTEM

The successful fulfillment of a space mission is dependent on the reliable functioning of the power system of the orbiting spacecraft. The stringent demands on performance, weight, volume, reliability and cost make the design of the spacecraft power system a truly challenging endeavor.

Significant advances have been made in this area resulting in the development of reliable and lightweight power systems for long duration missions (typically more than five years). Since a space mission is inherently expensive, the necessity of optimization and built in reliability becomes a rule rather than an exception for all on-board systems. Therefore, continuous efforts are being made to realize better performance from power systems.

Elements of a Spacecraft Power System

The amount of electrical power required on board a spacecraft is dictated by the mission goals (i.e. the nature and operational requirements of the payloads, the antenna characteristics, the data rate, the spacecraft orbit, etc.) Uninterrupted power must often be supplied for durations up to ten years or more.

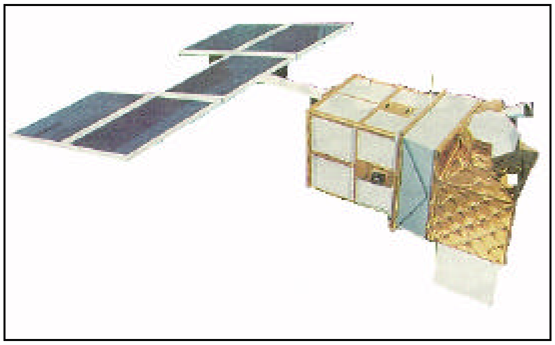

Fig. 6-6. Solar Panels provide energy for earth-orbiting satellites

The generation of electrical power on board a spacecraft generally involves four basic elements:

- A source of energy, such as direct solar radiation, nuclear power or chemical reactions
- A device for converting the energy into electrical energy
- A device for storing the electrical energy to meet peak and/or eclipse demands
- A system for conditioning, charging, discharging, regulating and distributing the generated electrical energy at specified voltage levels

The most favorable energy source for Earth-orbiting satellites is solar radiation (**Fig. 6-6**). Solar radiation bombards the Earth at a level of 126 watt/ft^2. Nearly all Earth-orbiting spacecraft use solar radiation as a source of energy. Because satellites pass into and out of the Earth's shadow solar radiation may not be useable by itself. A supplemental energy storage device must be used to provide power during eclipse and peak demand periods. Chemical sources such as rechargeable storage batteries serve this purpose. These batteries employ electrochemical processes and have typical efficiencies of 75%.

As an alternative to solar energy, radioactive isotope generators have also been used. This power source is especially practical for exploration missions to the outer planets where solar radiation levels are low. For example, the solar radiation reduces from about 54 watt/ft^2 in the vicinity of Mars to about 4.6 watt/ft^2 near Jupiter. It therefore becomes necessary to use other primary sources of energy for spacecraft missions to Jupiter and beyond.

Batteries and fuel cells produce electrical power through chemical reactions. The chemical dynamic system uses the heat energy liberated by some chemical reactions to heat a working fluid, such as sodium, and turn a generator. The chemical dynamic system is not considered practical for space usage and will not be addressed.

Photovoltaic and solar thermoionic devices both harness energy from the Sun. The photovoltaic energy source uses potential differences created by electromagnetic radiation illuminating semiconductors to provide power. The solar thermoionic system uses a temperature gradient set up across different types of semiconductors to create a flow of current. It is seldom used and thus, will not be discussed.

Another source is the solar dynamic system. This system can theoretically provide many kilowatts of power for extremely long mission durations. In this system, the energy from the Sun is focused onto a vessel containing a working fluid. The fluid heats, expands and can be used to turn a generator. This method will be employed on the U.S. Space Station to supplement its batteries and solar arrays.

Nuclear power uses the heat energy produced during nuclear fission to generate power.

Choosing a spacecraft power source for a particular mission may be difficult. Continuous power requirements, eclipse conditions as well as power subsystem weight are major factors in the final choice. Sometimes, a combination of energy sources is may be required.

Solar Arrays

Solar arrays are mounted on the satellite in various forms. They may be body mounted, stationary, or on directional, steerable wings. A solar array consists of solar cells which

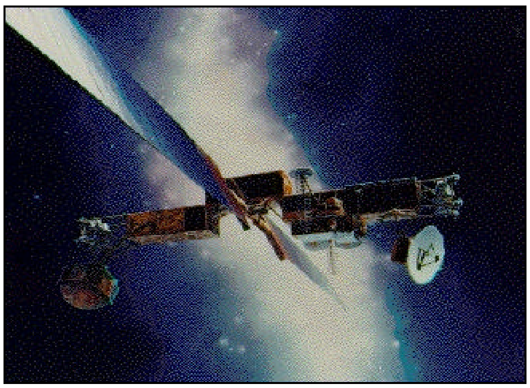

Fig. 6-7. Spacecraft with large solar array

convert solar energy into electric power by the photovoltaic effect. An array also contains interconnectors to connect the solar cells, panels on which the solar cells are mounted and mechanisms to deploy the panels in orbit (**Fig. 6-7**).

The power output of a single cell is quite low, so the individual cells are arranged in series to provide the desired voltage, and in parallel to render the desired current requirements. In addition, solar array modules are often constructed with several strings in parallel which are connected together in a series; a parallel "ladder" network. This is done to minimize power loss with a single cell failure. If each string were used independently, the loss of a single cell would create an open circuit for that entire string and the output from that string would be totally lost. With the ladder network arrangement, connectivity among the remaining cells is maintained in any string with a failed cell. However, the output may be somewhat degraded.

Some satellites cannot use deployable solar arrays because of the type of attitude control system they employ. Spin-stabilized satellites cannot support deployable solar arrays because of the stresses placed on the panels while the satellite rotates. For this reason, spin stabilized satellites require body mounted solar arrays. Body mounting is a very simple approach that utilizes available space on the satellite surface.

Some solar arrays are directional. A solar array drive is employed to control the angle of the arrays continuously so they may be always perpendicular to the sun's rays. In contrast, stationary arrays are deployable arrays locked into position relative to the spacecraft body once deployed.

The power of a solar array varies with time due to:

- The variation in solar intensity
- Variation in the angle between the solar array surface normal and solar rays
- Radiation degradation in solar cell power characteristics
- Array contamination by thruster propellants, etc.

The solar array power characteristics during the lifetime are obtained by superimposing the seasonal variation in power output on the time-varying radiation thermal degradation characteristic.

Solar array size is driven by a combination of satellite power requirements and the efficiency of the solar cells to convert solar energy to electrical energy. At the distance the Earth is from the sun (1 AU), the solar constant is 126 watt/ft^2. A new cell is only 11-12 % efficient when the Sun's rays are incident at 90° to the surface. As a result, solar cell output is about 10 watt/ft^2. To improve their efficiency and protect the cells from particles and radiation, the cells are covered with multiple layers of materials to protect them and enhance conversion. A silicon dioxide layer is used to enhance the desired wavelengths resulting in higher efficiency and giving the cells their characteristic blue color. In order to provide a kilowatt of power to the satellite payload and vehicle subsystems, solar array must have about 100 ft^2 of surface area.

Optimum solar array efficiency is usually achieved between 25° C and 28° C. The cell may be laminated with a material to reflect the 4,000 Angstrom wavelength radiation to keep the cell cooler. A change in cell temperature will change its voltage-current characteristics. An increase in the cell operational temperature causes a slight increase in the cell current and a significant decrease in the cell voltage. Therefore, the overall efficiency decreases as temperature increases. Thermal control of the solar array panels is achieved by the absorption of solar radiation by the solar cells on the front surfaces of the panels and reemission of infrared energy from the front and back of the panels.

The power from the solar cell will be maximum when the angle of incidence of illuminating light is zero (i.e. it is perpendicular to the solar cell surface). The power decreases as the angle of incidence deviates from zero. The primary reason for the increased loss of power at greater angles of incidence is the change in reflection coefficients at large angles.

Storage Batteries

In most spacecraft power systems that use solar radiation, the storage battery is the main source of continuous power. Batteries must provide continuous power to the spacecraft during peak power cycles and eclipse periods. The frequency and duration of eclipse periods depend on the spacecraft orbit.

The eclipse seasons in geostatioary orbits occur twice per year, during spring and autumn. These eclipse seasons are 45 days long and center around the vernal and autumnal equinoxes. There is one eclipse period per 24 hours with the maximum period of 72 minutes. The batteries discharge during an eclipse and are charged during the sunlight period. So, the charge-discharge cycles for any storage battery on board a spacecraft in geosynchronous orbit will be about 90 per year.

In the case of low orbiting satellites, the number of eclipses increases as the altitude of the satellite decreases. For a 550 km circular orbit there will be about 15 eclipses per day. The maximum shadow duration is about 36 minutes during each 96 minute orbit. There will be about 5,500 charge-discharge cycles per year in this orbit. Depending on the orbit inclination, the spacecraft may be in continuous sunlight for long periods several times a year

As mentioned above, batteries are necessary to maintain steady, reliable spacecraft power. A battery is an electrochemical device that stores energy in the chemical form and then converts it into electrical energy during discharge. Chemical reactions taking place inside the battery produce electrical energy whose magnitude is dependent upon various cell characteristics (i.e. individual cell voltage, efficiency of the electro-chemical reaction, size of the cell, etc.).

Batteries are classified as either primary or secondary. Primary batteries are used on spacecraft in which the battery is the only source of electrical power and it cannot be recharged. Thus, primary batteries are used for short duration missions of usually less than a week. Primary batteries have the advantages of being cheap, reliable and can deliver relatively large amounts of energy per pound of battery (20-100 watt-hr/lb).

Secondary batteries, are rechargeable. They convert chemical energy into electrical energy during discharge, and convert electrical back to chemical energy during recharge. This process can be repeated many times. Secondary batteries are used for longer duration missions such as Defense Meteorological Satellite Program (DMSP), Defense Satellite Communications System (DSCS) and many others, where solar arrays are the primary source of power.

The advantages of secondary batteries are:

- Capability of accepting and delivering unscheduled power at high rates (eclipse operations and peak power demands)
- Large number of charge-discharge cycles or long charge-discharge cycle life under a wide range of conditions
- Long operational lifespan
- Low volume
- Low cost
- High proven reliability

The disadvantages are:

- The memory effect process
- The complexity and expense of charge-discharge monitoring equipment
- Low energy storage capability per pound of battery (2-15 watt-hr/lb)

There are many types of secondary batteries available. However, only some are considered suitable for space applications.

The nickel-cadmium (Ni-Cad) battery is probably one of the most common batteries used in spacecraft today. It has four main components: the cadmium negative electrode, which supplies electrons to the external circuit when it is oxidizing during discharge; the nickel positive electrode, which accepts the electrons from the external circuit; the aqueous electrolyte, 35% KOH, which completes the circuit internally; and a separator made of nylon or polypropylene, which holds the electrolyte in place and isolates the positive and negative plates.

The primary factors affecting the useful life of a Ni-Cad cell are battery temperature, depth of discharge and excessive overcharge. The most important effect of high battery temperature is the reduction of separator life. Prolonged exposure of a Ni-Cad battery to high temperature will hasten the decomposition of the separator material. The repeated overcharging at low temperatures can result in pressure buildup. Therefore, battery temperature is an extremely critical parameter in the battery life design. It is common practice to use a thermal radiator to keep battery temperature below 24° C and to use heaters to keep it above 4° C.

Repeated deep discharges tend to degrade the cell plate structures, causing cracking. These cracks absorb electrolyte and gradually the separator dries out. For a synchronous orbit application of 7 to 10 years, a battery will encounter approximately 1,000 charge-discharge cycles over its lifetime. For this number of cycles, Ni-Cad battery depth of discharge is generally limited to 50 to 60%.

The batteries exhibit a gradual decay of terminal voltage during successive discharge periods. This effect is most pronounced when the charge-discharge cycle is repetitive, and is referred to as the “memory effect.” When the battery is cycled to a fixed depth of discharge, the active material that is not being used gradually becomes unavailable, resulting in an effective increase in depth of discharge. In addition to the gradual decay of discharge voltage, the batteries will also exhibit a tendency toward the divergence of the individual cell voltages during charge and discharge. Battery performance can be restored to a certain extent by reconditioning. A typical reconditioning process for a rechargeable battery consists of effecting a deep discharge and then recharging at a high rate. Reconditioning is a process begun a few weeks before eclipse season on many spacecraft.

Procedures to enhance battery life include maintaining batteries within a small temperature range, proper reconditioning, and trickle charging between eclipse seasons, to prevent cadmium migration from negative electrodes to positive electrodes.

Another type of secondary battery is the nickel-hydrogen battery (Ni-H_2). This battery is actually hybrid battery-fuel cell device. It has a positive electrode, much like a conventional battery and a fuel cell negative electrode. Hydrogen gas is diffused onto a catalyst, usually platinum, at the negative electrode where the reaction occurs. High pressure vessels (500 psi) are required to contain the gas.

Nickel-hydrogen batteries are increasingly being used on newer spacecraft such as MILSTAR and replacement GPS satellites. Compared to Ni-Cad batteries, NiH_2 have higher specific energy, can tolerate a higher number of discharge-recharge cycles and operate at near-optimum output over a wider range of temperatures.

Fuel Cells

Fuel cells have played a major role in the NASA manned spaceflight programs. They were originally chosen over batteries as primary electrical power sources for these applications because conventional batteries could not meet the energy density requirements for the Gemini and Apollo space missions. Consequently, the decision was made to develop fuel cells for manned orbital flights. The fuel cells were to be powered by cryogenic hydrogen and oxygen stored in pressurized insulated tanks.

A fuel cell is a device that directly converts the chemical energy of reactants (a fuel and oxidizer) into low voltage direct current (DC) electricity. Like the primary and secondary batteries discussed earlier, a fuel cell accomplishes this conversion via electrochemical reactions. However, unlike these conventional batteries, it does not consume reactants that are stored within its structures, but uses reactants that are stored in external tanks. A fuel cell consists of two sintered, porous nickel electrodes separated by an ion-conducting electrolyte like potassium hydroxide. The fuel cell can operate as long as it is continuously fed with reactants and reaction products are removed. Because it is easy to make the reactant tanks larger, a fuel cell's period of operation can be made much longer than that for a conventional electrochemical battery.

At the negative electrode, incoming hydrogen gas ionizes to produce hydrogen ions and electrons. Since the electrolyte is a non-electronic conductor, the electrons flow away from the electrode into the external circuit. At the positive electrode, oxygen gas reacts with migrating hydrogen ions from the electrolyte and incoming electrons from the external circuit to produce water. Depending on the operating temperature of the fuel cell, the product, water, may enter the electrolyte, thereby diluting it or be lost as vapor through the cathode. In fuel cells with liquid electrolytes that operate below the boiling point of water, an electrolyte circulation system incorporating an external evaporator may be necessary to remove the water.

In any event, as long as hydrogen and oxygen are continuously fed to the fuel cell, the flow of electric current will be sustained. By electrically connecting a number of cells together in series or parallel, it is possible to form a fuel cell "stack" of any desired voltage or current. The U.S. Space Shuttle uses three fuel cells, each with an output of approximately 30-33 V at 250 A, to produce power for various loads during a mission. Since fuel cells produce water as a by-product, fuel cells provide potable drinking water for the crew and can be used as evaporator cooling for the vehicle.

Among the primary advantages of fuel cells are that they provide continuous power (as long as fuel and oxidizer are supplied), have low weight but high output power and produce water as a by-product. Their advantages make them very useful for manned missions.

The main disadvantages of fuel cells are their expense and the possibility that loss of cooling can result in an explosion. Consequently, elaborate control systems are required to keep them operating.

Nuclear Power

Most spacecraft nuclear power generators are capable of delivering a range of power from a few watts up to several hundred. The challenging problems encountered with shielding nuclear power generators have precluded their use on manned missions **(Fig. 6-8)**. However, they have been used very successfully on many deep space missions where solar flux levels are too low for photovoltaic solar cells to be effective.

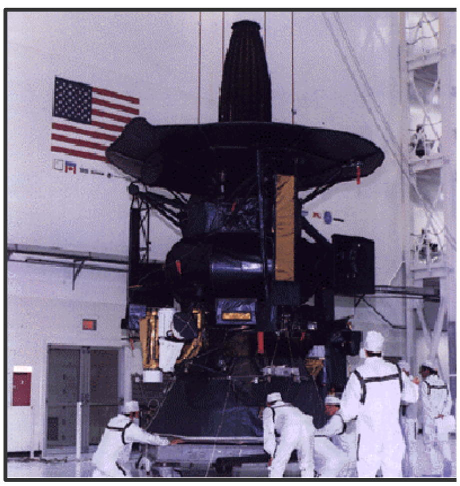

Fig. 6-8 NASA Gallileo Spacecraft with two nuclear powered generators

Political and environmental problems with nuclear powered satellites were underscored in 1978 after COSMOS 954 plunged to earth, scattering nuclear material over a large part of northwest Canada. From the beginning of the U.S. space nuclear power program, great emphasis has been placed on the safety of people and the protection of the environment. The operational philosophy adopted for orbital missions requires that the normal lifetime in space be long enough to permit radioactive decay of the radioisotope fuel to a safe level prior to reentry into the Earth's biosphere. Stringent design and operational measures are used to minimize the potential interactions of the radioactive materials with the global populace and to keep any such exposure levels within limits established by international standards.

Like fuel cells, nuclear power generators have a major role in space exploration. There are two basic types of nuclear powered generators. Radioisotope thermoelectric generators (RTG) rely on the decay of radioisotopes. The second type uses the heat of the nuclear fission process, much like nuclear generators on earth. Both processes involve material having high energy radiation levels of several types: alpha particles, beta particles and gamma particles.

This energy can be harnessed to produce electricity on spacecraft. The radioactive material is encased in a special metal container from which the decay particles cannot escape. As the container absorbs energy produced by the alpha and beta particles, it is heated to a high temperature. This heat can be employed in conjunction with a thermoelectric couple to produce the necessary electricity. The heat from a nuclear reactor can be utilized in two basic methods called static and dynamic. The static method uses no moving parts and is usually preferred for this reason. The dynamic conversion systems use the heat to perform mechanical work on a turboalternator assembly which generates the electricity.

The advantages of nuclear energy include its ability to provide power for long duration missions without reliance on solar illumination, high system reliability and high power output versus low mass.

Among the primary disadvantages of nuclear power systems are their high cost, heavy shielding requirements (which restricts their use on manned missions), need for complex cooling systems to prevent core meltdown and relatively low efficiencies (less than 18% efficiency). The high level of environmental concern and corresponding political ramifications all but preclude use of nuclear systems in Earth orbiting satellites.

ATTITUDE CONTROL SUBSYSTEM

Attitude control can be defined as the process of achieving and maintaining a desired orientation in space. An attitude maneuver is the process of reorienting the spacecraft from one attitude to another. An attitude maneuver in which the initial attitude is unknown, when maneuver planning is being undertaken, is known as attitude acquisition. Attitude stabilization is the process of maintaining an existing attitude relative to some external reference frame. This reference frame may be either inertially fixed or slowly rotating, as in the case of Earth orbiting satellites.

An attitude control system is both the process and hardware by which the attitude is controlled. In general, an attitude control system consists of three components: navigation sensors, guidance section and control section. A navigation sensor locates known reference targets such as the Earth or Sun to determine the spacecraft attitude. The guidance section determines when control is required, what torques are needed and how to generate them. The control section includes hardware and actuators that supply the control torques.

Definitions

Station Keeping—The sequence of maneuvers that maintains a vehicle in a predetermined orbit

Attitude—The position or orientation of a body, either in motion or at rest, as determined by the relationship between its axes and some reference line or plane such as the horizon

Attitude Adjustment—Changing the orientation of the spacecraft within its orbit

Orbit Adjustment—Changing the orbit itself

Navigation—Determination of spacecraft's current position and velocity

Guidance—Computation of corrective actions

Control—Implementation of corrective actions

Stabilization—The property of a body to maintain its attitude or to resist displacement, and, if displaced, to develop forces and movements tending to restore the original condition

Perturbation—A disturbance in the regular motion of a celestial body, the result of a force additional to that which caused the regular motion, specifically, a gravitational force.

Coordinate System—Any scheme for the unique identification of each point of a given continuum. Various systems in use are Polar, Cartesian, Spherical, and Celestial

Active and Passive Control Systems

There are two categories of attitude control systems: active and passive. Active systems use continuous decision making and hardware (closed loop) to maintain the attitude. The most common sources of torque actuators for active control systems are thrusters, electro-magnets and reaction wheels. In contrast, passive attitude control makes use of environmental torques (open loop) to maintain the spacecraft orientation.

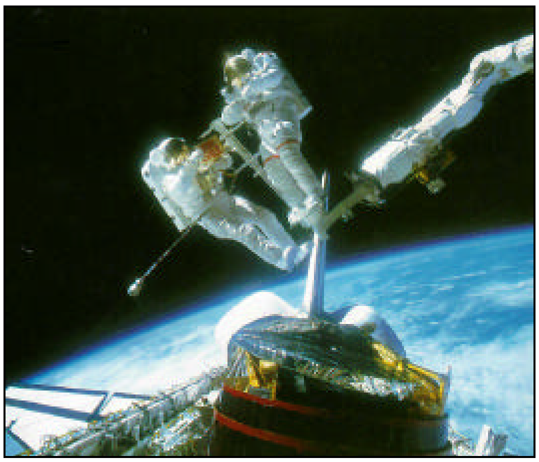

Fig. 6-9. Astronauts working on satellite with a gravity boom

Gravity gradient and solar sails are common passive attitude control methods (**Fig. 6-9**).

Attitude control systems are highly mission dependent. The decision to use a passive or active control system or a combination of the two depends on mission pointing and stability requirements, mission orbital characteristics and the control system's stability and response time. For example, a near-Earth, spin-stabilized spacecraft could use magnetic coils for attitude maneuvers and for periodic adjustment of the spin rate and attitude. Above synchronous altitudes, thrusters would be required for these functions because the Earth's magnetic field is generally too weak at this altitude for effective magnetic maneuvers.

Any satellite orbit requires stabilization to increase its usefulness and effectiveness. For instance, when a satellite is not stabilized, it must use omni-directional antennas so that ground stations can receive its downlink information regardless of the satellite's orientation. This necessitates a high power transmitter and only a small portion of the total power is radiated to Earth. On the other hand, if there are means to stabilize the satellite so its directional antennas can be pointed at the Earth, then lower power may be used to transmit information to the ground.

There are four functions spacecraft attitude control systems incorporate: satellite pointing, orbital transfer maneuvers, stabilization against torques and satellite de-spin. Solar arrays generate maximum power when they are perpendicular to the Sun. In addition, some satellites carry scientific payloads which must observe a celestial body. In order to observe it, the spacecraft must be able to accurately find the object, track it and point applicable sensors at it. Sensors must be accurately pointed at Earth to detect ICBM launches as well as movement of troops, ships, aircraft, etc.

During orbital transfer maneuvers, it is necessary to be as precise as possible. Therefore, stringent requirements on the accuracy of the spacecraft orientation must be achieved by the attitude control system before firing. Aligning the spacecraft for perigee and apogee motor firing requires a knowledge of the orbit characteristics at the time at which the motors are fired. This knowledge optimizes the transfer maneuver by minimizing both time and propellant requirements for the orbital transfer. If the spacecraft relies on solar energy for electrical power generation during the transfer maneuver, then the spacecraft must be optimized for maximum solar cell illumination during the transfer. Finally, the spacecraft must be reoriented again after the completion of the transfer maneuver.

Disturbance torques are environmental torques (i.e. drag, solar wind, magnetic field, gravity, micrometeoroid impacts, center of gravity changes) or unintentional internal torques (i.e., liquid propellant slosh). Because these can never be totally eliminated, some form of attitude control system is required. Control torques, such as those produced by thrusters, are generated intentionally to control spacecraft attitude.

Traditionally, spacecraft employing solid propellant apogee motors have adopted spin-stabilization during the parking and transfer phases. Even spacecraft that have active attitude control systems in their operational orbits are frequently spin-stabilized in an initial (transfer orbit) phase of their mission. Spin stabilization during transfer orbit allows thermal control to be distributed evenly throughout the spacecraft. If the spacecraft is required to be three-axis stabilized, it must be despun before being injected into the appropriate attitude. If the spacecraft is to be spin stabilized then the spin rate must be increased or decreased, depending on the final spin rate required.

Navigation Sensors

As mentioned before, sensors are required to determine the orientation of the spacecraft and its current state. The types of sensors used on a particular vehicle depend on several factors including the type of spacecraft stabilization, orbital parameters, operational procedures and required accuracy.

Sun sensors are the most widely used sensor types; one or more varieties have flown on nearly every satellite. The Sun is sufficiently bright to permit the use of simple, reliable equipment without discriminating among sources and with minimal power requirements. Many missions have solar experiments, most with Sun-related thermal constraints, and nearly all require the Sun for power. Consequently, missions are concerned with the orientation and time evolution of the Sun vector in body coordinates. Attitude control systems are frequently based on the use of a Sun reference pulse for thruster firings, or more generally, whenever phase-angle information is required. Sun sensors are also used to protect sensitive equipment, such as star trackers, from harmful particle bombardment as well as to position solar arrays to achieve maximum power conversion efficiency.

The orientation of the spacecraft to the Earth is of obvious importance to space navigation, communications, weather and Earth resources satellites. To a near-Earth satellite, the Earth is the second brightest object and covers up to 40% of the sky. The Earth presents an extended target to a sensor compared with a point source approximations used for Sun and star detectors. Consequently, detecting only the presence of the Earth is normally insufficient for even crude attitude determination and nearly all sensors are designed to locate the Earth's horizon .

Unfortunately, the location of the Earth's horizon is difficult to define because its atmosphere causes a gradual decrease in radiated intensity away from the true or hard horizon of the solid surface. Earth resources satellites such as Landsat **(Fig. 6-10),** communications and weather satellites typically require a pointing accuracy of 0.05 degrees to less than a minute of arc, which is typically beyond the state of the art for horizon sensors.

Earth emanates infrared radiation, and the IR intensity in the 15 micron spectral band is relatively constant. Most horizon sensors now use the narrow 14 to 16 micron bands. Use of the infrared spectral band avoids large attitude errors due to spurious triggering of visible light horizon sensors off high altitude clouds. In addition, the operation of an infrared horizon sensor is unaffected by night. Infrared detectors are less susceptible to sunlight reflected by the spacecraft than are visible light detectors and therefore, avoid reflective problems. Sun interference problems are also reduced in the infrared band where the solar intensity is only 400 times that of the Earth, compared with 30,000 in the visible spectrum.

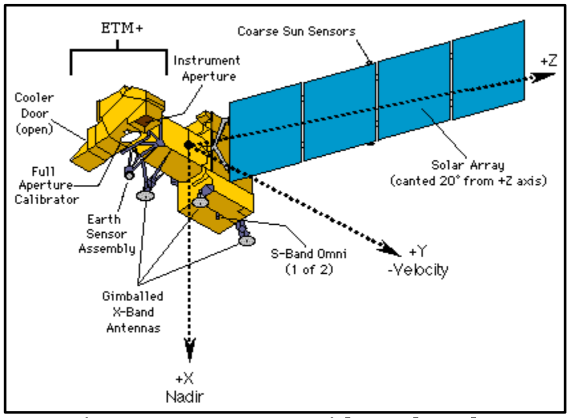

Fig. 6-10 LANDSAT with earth and star sensors

Most horizon sensors consist of four basic components: a scanning mechanism, an optical system, a radiance detector and signal processing electronics. The scanning mechanism is used to scan the celestial sphere and houses the actual sensor.

The optical system of a horizon sensor consists of a filter to limit the observed spectral band and a lens to focus the target image on the radiance detector. Optical system components depend greatly on the sensor design. In many cases, rotating mirrors or prisms are incorporated into the optical system to provide the scanning mechanism.

Knowledge of the scan rate or duty cycle allows the conversion from time to angle either on board the satellite or on the ground. Typically, sensors are designed and calibrated so that the system output may be used directly for attitude control and determination within a specified accuracy under normal operating conditions.

Some horizon sensor systems have been designed for specific mission conditions and thereby achieve increased accuracy and simplicity but at the cost of reduced versatility. These systems operate over a narrow range of orbits and attitudes and include moving and static edge trackers and radiometric balance systems.

Star sensors measure star coordinates in the spacecraft frame and provide attitude information when these observed coordinates are compared with known star positions and magnitudes obtained from a star catalog. In general, star sensors are the most accurate of navigation sensors, achieving accuracy to the arc-second range. However, this capability is not achieved without considerable cost. Star sensors are heavy, expensive and require more power than most other navigation sensors. In addition, computer software requirements are extensive, because measurements must be preprocessed and identified before attitudes can be calculated. Because of their sensitivity, star sensors are subject to interference from the Sun, Earth and other bright objects. In spite of these disadvantages, the accuracy and versatility of star sensors have led to applications in a variety of different spacecraft attitude control systems.

Star sensing and tracking devices can be divided into three major categories: star scanners, which use the spacecraft rotation to provide searching and sensing function; gimbaled star trackers, which search out and acquire stars using mechanical action; and fixed head star trackers, which have electronic searching and tracking capabilities over a limited field-of-view. Sensors in each of these classes usually consist of the following components: a sun shade; an optical system; an image definition device which defines the region of the field of view that is visible to the detector; the detector; and an electronics assembly. Furthermore, gimbaled star trackers have gimbaled mounts for angular positioning.

Stray light is a major problem for star sensors. Therefore, an effective sun shade is critical to star sensor performance. Carefully designed light baffles are usually employed to minimize exposure of the optical system to sunlight and light scattering caused by dust particles, clouds and portions of the spacecraft itself. Even with a well designed sun shade, star sensors are typically inoperable within 30 to 60 degrees of the Sun.

The star sensor optical system consists of a lens which projects an image of the star field onto a focal plane. The image definition device selects a portion of the star field image in the sensor's field of view which will be visible to the detector. This portion is known as the instantaneous field of view (IFOV). The image definition device may be either a reticle consisting of one or more transparent slits etched on an opaque plate, or an image dissector tube in which the IFOV electronically scans the FOV. The detector transforms the optical signal into an electronic signal. Finally, the electronics assembly filters the amplified signal received from the detector and performs many functions specific to the particular star sensor. These functions include defining the magnitude of the stars as well as relative positions which will be used to determine spacecraft attitude.

Star scanners used on spinning spacecraft are the simplest of all star scanners because they have no moving parts. The image definition device employed by this type of sensor consists of two V-slits through which the star light passes. The spacecraft rotation causes the sensor to scan the celestial sphere. As the star image on the focal plane passes a slit, the star is sensed by the detector. If the amplified optical signal passed from the detector to the electronics assembly is above a threshold value, then a pulse is generated by the electronics signifying the star's presence. The interpretation of the star scanner measurements becomes increasingly more difficult as spacecraft motion deviates from a uniformly spinning body.

Gimbaled star trackers are commonly used when the spacecraft must operate at a variety of attitudes. This type of tracker has a very small optical field of view (usually less than one degree). Gimbaled star trackers normally operate on a relatively small number of target stars. A major disadvantage of gimbaled star trackers is that the mechanical action of the gimbal reduces their long term reliability. In addition, the gimbal mount assembly is frequently large and heavy.

Fixed head trackers use an electronic scan to search their field of view and acquire stars. They are generally smaller and lighter than gimbaled star trackers and have no moving parts.

Magnetometers can be used to measure both the direction and magnitude of the Earth's magnetic field to the milligauss accuracy. They are reliable, lightweight and have low power requirements. They operate over wide temperature range and have no moving parts. However, magnetometers are not accurate inertial navigation sensors because the Earth's magnetic field is not completely known and the models used to predict the magnetic field direction and magnitude at the spacecraft's position are subject to relatively substantial errors. Furthermore, because the Earth's magnetic field strength decreases with distance from the Earth (Earth's magnetic field in low Earth orbit is about 0.5 gauss) residual spacecraft magnetic biases eventually dominate the total magnetic field measurement. Magnetometers are generally limited for use to spacecraft with altitudes below 1,000 km.

A gyroscope is any instrument which uses a rapidly spinning mass to sense and respond to changes in the inertial orientation or its spin axis. There are three basic types of gyroscopes used on spacecraft: rate gyros, rate-integrating gyros, and control moment gyros. The first two types are attitude sensors used to measure changes in the spacecraft orientation. Control moment gyros generate control torques to change and maintain the spacecraft's orientation.

Rate gyros measure spacecraft angular rates and are frequently part of a feedback system for spin rate control or attitude stabilization. The angular rate outputs from rate gyros may also be integrated by an on-board computer to provide an estimate of spacecraft attitude displacement from some initial reference. Rate-integrating gyros measure spacecraft angular displacement directly. In some applications, rate-integrating gyro output consists of the total spacecraft rotation during small time intervals. An accurate measure of the total attitude displacement may then be obtained by integrating the average angular rates constructed from incremental displacements.

Control Actuators

Control actuators are used to correct the attitude of a spacecraft such that it attains and stays in the desired attitude. There are two types of attitude control methods: passive and active.

Passive Attitude Control

Passive attitude control techniques include spin stabilization and gravity gradient stabilization. Passive systems involve no active elements, require no altitude sensors and have a high reliability. However, passive systems are extremely sensitive to environmental torques and payload motion, limited to near circular orbits, and have a local vertical accuracy from 2 to 3 degrees at 500 miles to 10 degrees at synchronous altitude.

Spin-stabilization is a passive technique that involves spinning the vehicle is at a constant rate (on the order of once per second). Because of their motion, spinning satellites can only scan a target rather than fix on a point . Thus “dual-spin” satellites were developed in which part of the satellite is spun for stabilization, while another part is nonspinning ("despun") for mission requirements such as pointing an antenna. The spinning motion gives angular momentum to the vehicle, which tends to reduce the effects of small disturbance torques on vehicle orientation.

Gravity gradient stabilization works by orienting the spacecraft along an axis of gravitational force. Gravitational influences can be significant enough at low altitude that a satellite can maintain fairly stable orientation. The gravity gradient approach takes advantage of gravitation's stronger pull closer to the Earth than when farther away. The difference between the close and far conditions can be used to generate a torque which aligns the satellite long axis with the local vertical. One method is to place the vehicle such that the maximum moment of inertia (usually the longest dimension) aligns with the local vertical pointing towards or away from the Earth. If a disturbance alters the vehicle out of this orientation, the varying force of gravity acting on different parts of the vehicle returns it to its original stable orientation. This effect can be greatly enhanced by extending lightweight booms with small weights on the ends. The boom attempts to align with the vertical, but will oscillate about the vertical, unless some damping mechanism is employed. In order to stiffen the damping force, horizontal biasing booms or high gain electrical nulling devices are used. Passively damped systems are favored for small satellites (less than 1,000 pounds) and for low altitudes (below 1,000 nautical miles), where the gravity gradient torque is stronger.

Active Attitude Control

Active attitude control techniques include momentum exchange, mass expulsion, external magnetic torques and solar torques. Momentum control devices are the most common attitude control actuators. They work by varying the angular momentum of small masses within the spacecraft. There are three types of these devices: momentum wheels, reaction wheels and control-moment gyroscopes.

Momentum wheels are the simplest, and consist of a single, constantly spinning wheel. The rotating mass gives the satellite a stiffness or resistance to outside torques. To provide pointing in more than one axis, momentum wheels are oriented at 90° angles to each other. Thus, the satellites orientation can be fixed within 0.1 and 10°.

Reaction wheels differ from momentum wheels in that they spin only when the satellite needs to be oriented differently or when controlling the effect of an externally produced torque. Spacecraft employing these devices usually have three or four, oriented at right angles. The fourth is for redundancy. Reaction wheels can achieve 0.001° pointing accuracy.

Both the momentum and reaction wheel devices have limiting speeds. If disturbances act continuously in the same direction so that wheel speed approaches maximum (saturation), the spacecraft must use another method (like mass expulsion) to reduce or "dump" angular momentum. To change a vehicle’s orientation, the motor can be commanded to change the spin rate of the wheel. The vehicle compensates in the opposite direction in order to conserve momentum.

Control-moment gyroscopes are the other momentum-based active control devices. They are as accurate as reaction wheels but can respond at a faster rate, making them more desirable in tracking applications. They are also more expensive and complex than reaction wheels. Mass expulsion uses a propulsion system to perform both small velocity corrections and attitude control. A large number of small thrusters, called reaction-control jets, can work together to provide translational acceleration (velocity correction) or can work in pairs to provide torques for attitude control. For angular motion, the thrusters are located near the vehicle extremities in order to develop the maximum torque for the least thrust or thruster size. Thrusters operate by expelling either hot or cold gas. Cold gas is stored under pressure and released to provide thrust. Hydrozine is a hot gas system using a chemical catalyst instead of an oxidizer for combustion. The rate of mass expenditure, or the total mass expended, depends upon the angular or linear velocity required, the size of the vehicle and the location of the thrusters. This active control system is insensitive to disturbance torques, provides the widest variety of control orientations and is highly precise. The major drawback in any mass propulsion system is the need to carry propellant, which adds considerable weight to the vehicle, especially for long missions.

Magnetic and solar torquers make use of environmental forces to impart stability and develop attitude changes. Electrical current run around a piece of metal on the spacecraft creates an electromagnet which will align itself along the Earth's magnetic field. These magnetic controls are relatively light and can be programmed to desaturate the momentum wheels. They can also be used to compensate for the natural magnetic effects of satellite components.

Torques caused by solar radiation pressure can be used for attitude control by orienting panels in the flow of solar radiation. The torques are small, but when extended over a long period, can have significant affect. Advantages to solar and magnetic torquers are that they do not require onboard propellant and they provide smooth corrections. On GPS satellites, magnetic torquing is the primary method of reaction wheel desaturation. Solar torques provided attitude control on the Mariner IV spacecraft.

TELEMETRY, TRACKING AND COMMANDING SUBSYSTEM (TT&C)

Telemetry

Telemetry is measurement data transmitted to operators at ground stations over a radio link. It contains information which is used to evaluate both the satellite's and the booster's performance. Telemetry data transmissions begin prior to launch, and continue throughout the life of the satellite. Launch and injection into orbit are especially critical times because data from the booster, upper stages (if used) and satellite must be received and evaluated.

The satellite's telemetry data, whether analog or digital, contains two general classes of information. The first, payload or mission data, varies with the mission of the satellite. Examples of payload data types are meteorological, oceanographic, astronomical and Earth resources information. Spacecraft health and status data types are relatively standard, regardless of the type of mission. This data consists of pressure, temperatures, flow rate, current, voltages and events as they occur throughout the satellite systems, subsystems and components. Once the satellite is on orbit, operators settle into more or less routine operations in which they continuously monitor the bus and payload telemetry in order to respond quickly to problems.

Tracking

Before we can communicate with a ballistic missile or orbiting satellite, we must know where it is with respect to our ground stations. Tracking is the process of making observations of the spacecraft's position relative to a tracking station or other fixed point whose position is accurately known. Orbit determination is a process in which the tracking observations are used to determine the spacecraft's orbital characteristics and its position in space. Tracking stations use elevation, azimuth, range and range rate data to determine satellite position relative to time.

The simplest way to track a satellite is through the use of a beacon or a transponder which announces the satellite's presence. In current tracking arrangements, most satellites use a transponder system from which range rate data is extracted. Beacons are also used as locating devices on recovery capsules. A transponder is triggered "On" only after receiving a specially coded signal from a ground tracking station. Upon receipt of this signal, the transmitter is activated and the coded signal is turned around by satellite and sent back to the tracking station. In addition to providing a greater degree of security, coded transponders enable operators in the tracking system to derive extremely accurate range information by measuring the elapsed time between the transmission and subsequent reception of the coded signal. Various other tracking methods include Doppler tracking, radar tracking and ranging, interferometer tracking and optical tracking.

Radar tracking satellites is similar to radar tracking aircraft. Radiofrequency energy is transmitted from a ground antenna up to the satellite. The energy reflected back to the ground is received by the tracking station. Satellite range is then computed by measuring the time required for the signal to make the trip. These measurements and calculations, taken over time, are used to determine range rate and relative motion.

Doppler tracking measures changes in transponder frequencies (satellite radio wave transmissions) which are caused by relative velocity differences. It requires calculation of the relative velocity of a satellite in relation to an observer (ground station).

Interferometer tracking measures phase differences in signals from the spacecraft, as received on the ground by precisely located antennas and reflectors.

As the name implies, optical tracking employs telescopes and optical instruments to search for and track the satellite via light reflected off its surfaces. This method can only be used at night and under clear skies.

Commanding

Commanding is the process of communicating to the satellite from a ground site (**Fig. 6-11**). The satellite is controlled via commands sent to change voltages, temperatures, aperture settings and other parameters aboard the spacecraft. This task is accomplished by transmitting coded instructions from the ground station over radio frequency carrier, referred to as the uplink, to the satellite's receiving equipment.

Examples of events executed by commanding include ascent control, orbit adjust, reentry by separation, engine ignition or cutoff and on/off of internal systems. In some cases, an entire sequence of events may be started by a single, preprogrammed command.

SINGLE commands are employed when controlling specific satellite functions. A SINGLE command is a command that is equal to one set of binary digits which will cause only one function to be performed in the satellite. A BLOCK command is one in which one command number may represent a number of single commands which will be transmitted to the satellite in a specific order.

Commands can be further identified as either Real-time Commands (RTC) or Stored Programs Commands (SPC). The primary difference between these commands is the time of execution. A Real Time Command initiates events on the satellite upon receipt of the command. RTC's are desired if command execution is necessary while the satellite is still within sight of a ground station. SPC's are sent to the satellite while it is still within a ground station view, but it causes certain functions to be performed after the satellite has passed out of sight of a ground station.

Some spacecraft have self-sustaining reference packages which contain preloaded commands. The advantage of such systems is that the preloaded commands allow the satellite to respond autonomously to situations and changes which are within an expected range of values. Spacecraft which contain no self-sustaining reference package must be continuously monitored and commanded by the ground control site for proper station keeping.

Fig. 6-11. A Ground Command and Control Site

TOC

REFERENCES

Air University Space Handbook (AU-18), A Warfighter's Guide to Space, Vol. II, 1993.

Bates, R.R., D.D. Mueller, and J.E. White. *Fundamentals of Astrodynamics.* New York, Dover Publications, Inc., 1994.

Pisacane, Vincent L. and Robert L. Moore, eds, *Fundamentals of Space Systems.* New York, Oxford University Press, 1994.

Sellers, Jerry Jon et. al., *Understanding Space, An Introduction to Astronautics.* New York, McGraw-Hill, Inc., 1994.

U.S. Army Space Reference Text, Space Division, U.S. Army Training and Doctrine Command, 1993.